Electrical Propulsion Requirements for Planetary and Interplanetary Spacecraft

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CONTENTS

FIGURES

ABSTRACT

/ 8 2 / 6   A

Assumptions are made concerning the nature of scientific payloads
required for planetary exploration missions and the minimum weight
of spacecraft necessary to deliver a scientific payload to its destination
in deep space. Although it is expected that early scientific exploration
of the Moon and of Venus and Mars will be carried out most eco-
nomically with chemically propelled spacecraft, it is suggested that
scientific exploration of the remainder of the solar system could be ac-
complished most economically through the use of electrical propulsion.

With relatively small variations in propellant loading, a spacecraft
of single basic design could perform any one of at least seven planetary
and two interplanetary scientific missions beyond Venus and Mars.
This spacecraft can use a two-stage *Saturn I-B* as a launch vehicle. The
parameters for such a mission program are discussed, and the power-
plant is defined. Finally, ultimate goals for a second-generation nuclear-
electric spacecraft are anticipated, and basic scheduling criteria are
established.   *Author*

I. INTRODUCTION

Under a prime contract to the National Aeronautics
and Space Administration (NASA), the Jet Propulsion
Laboratory (JPL) has been assigned the responsibility
for and is presently engaged in the unmanned scientific
exploration of the Moon and planets. The economic and
timely direction of that responsibility requires the use
of nuclear-electric power and electrical propulsion sys-
tems; the utilization of these systems is included in the
JPL–NASA long-range plan. Beyond Venus and Mars,
electrical propulsion offers the most realistic means of
mission accomplishment.

Planetary mission capabilities are probably best sum-
marized in reports by E. W. Speiser (Ref. 1) and T. N.
Edelbaum (Ref. 2), based on trajectory studies by W. G.
Melbourne (Ref. 3) and others (Ref. 4 and 5). Table 1
presents a partial summary of the mission potential for
electrical propulsion systems. The ratio of terminal mass
to initial mass (terminal mass ratio) may be increased by
a corresponding increase in power or flight time. The
reference studies are almost completely parametric in
that power and flight time are dependent on booster and
nuclear-electric powerplant capabilities. In order to de-

fine these two variables parametrically, the specific power level P_0^*, which is defined as kilowatts of electrical power delivered to propulsion per ton of spacecraft initial mass, is introduced. As shown in the referenced material (Ref. 1), a nuclear-electric powerplant with a specific weight of 20 lb/kwe would be optimum for a spacecraft specific power of 25 to 30 kwe/ton.

Since the manned lunar program has become the top priority program for the nation, funds available for the planetary space program are restricted. It follows, then, that the initial planetary program must be accomplished with a minimum of cost and effort, an economical approach that is inherently healthy in a technical program of the enormous scope anticipated. This Report illustrates conclusively that an electrically propelled spacecraft of a single, modest design, mounted on the *Saturn I-B*, a booster already under development and scheduled for early completion, can deliver more than 2 tons of net spacecraft anywhere within the solar system. By present standards this capability is significant. In addition, the use of a single modifiable spacecraft design represents the ultimate economy.

Table 1. Planetary mission capabilities for electrically propelled spacecraft

Mission	Terminal mass ratio	Specific power kwe/ton	Flight time days
Minor planet probe			
Mars probe	0.80	25	240
Venus probe	0.80	25	190
Mercury probe	0.80	25	270
Mars/Venus capture			
Mars capture	0.78	25	290
Venus capture	0.78	25	240
Major planet probe			
Jupiter probe	0.75	25	630
Saturn probe	0.75	25	930
Pluto probe	0.75	30	2000
	0.65	30	1650
Major planet capture			
Jupiter capture	0.70	30	750
Saturn capture	0.70	30	1300
	0.60	30	1090
Mercury capture			
Mercury capture	0.70	30	290

II. SCIENTIFIC PAYLOAD REQUIREMENTS

Parallel to the development of nuclear-electric propelled spacecraft is the development of the scientific payload that the craft will carry. Scientific instrument packages presently carried on space missions normally weigh just a few pounds. Assuming that hundreds of pounds are potentially available, a question arises concerning the best return of data for the least investment of money, manpower, and time.

Since June 1962, an advanced planetary spacecraft study committee at JPL has been investigating the question of "optimum payload" for planetary missions. Although specific requirements may vary from mission to mission, one general conclusion appears to hold: for first flights, at least 300 lb of actual scientific instruments can be efficiently used in the accumulation of exploratory data in planetary and interplanetary space. Of course, the weight of the equipment required for adequate positioning and orientation has not been included.

Until the initial exploratory work is accomplished, it will be difficult to determine what further instrumentation is needed. It is possible that a few vehicles with 300 to 500 lb of instruments may be much more valuable than a single spacecraft with some thousands of pounds of instruments. Undoubtedly, though, it is expected that greater payload capabilities will ultimately be desirable for certain types of planetary exploration, while smaller, more versatile capabilities may be desired in other applications.

Some of the pertinent mission criteria being projected for a Mars orbiter scientific package of 300 lb are essentially as follows:

1. Orbital requirements: minimum altitude, circular, near-polar orbit, with adequate coverage of daylight area for lifetime periods of at least 140 days and 100 orbits;

2. Spacecraft orientation: capability of orienting to a planet's vertical and inertial coordinates within 1 deg;

3. Power consumption: 400 w, with peaks to 10 kw;

4. Information capacity: at least 10^{10} total information bits coordinated with altitude information to 1 km and with surface position to 10 km.

For a Mars planetary lander, a completely different set of experiments has been proposed. However, the general spacecraft interface requirements do not change significantly. Apparently desired is a 350-lb, 250-w set of instruments with a lifetime of approximately one-half Martian year and a total information capacity of 5×10^9 bits. Mechanisms, optical equipment, etc., would, of course, require adequate positioning and orientation.

III. MULTIPLE COMPARISONS OF PROPULSION SYSTEMS

In order to demonstrate the reason for the selection of electrical propulsion for use in planetary missions, it is necessary to make planetary mission comparisons of the propulsion systems under development. The electric-propulsion mission expectations have already been shown in Table 1. For chemical-propulsion and nuclear-rocket systems, it is appropriate to discuss the planetary missions in terms of the total velocity increment ΔV. This mission summary is shown in Fig. 1 (Ref. 6), where ΔV_1 is the velocity increment required to send a spacecraft from an initial, low Earth orbit to achieve a flyby (probe), and ΔV_2 is the additional velocity increment required for the craft to achieve a capture orbit at the destination planet.
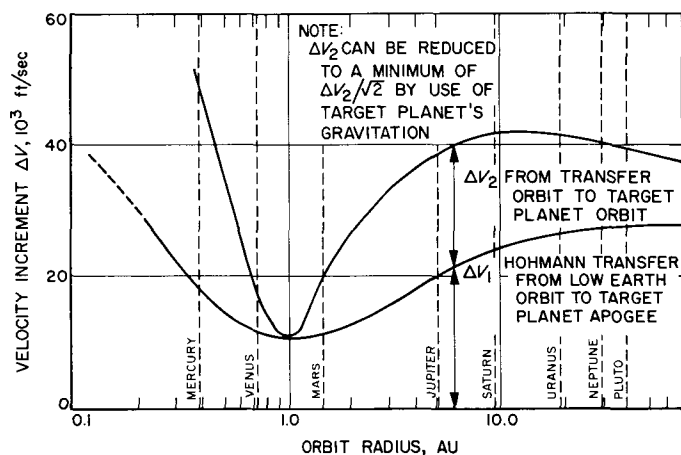

Fig. 1. Velocity requirements for interplanetary transfer missions

From Earth distance, where an added velocity increment of 10,000 ft/sec will allow a spacecraft to escape from the Earth's gravitational attraction, a Hohmann transfer closer to the Sun or farther away from the Sun requires a continually increasing velocity increment, ΔV_1 (see Fig. 1). The additional velocity increment, ΔV_2, needed for a planetary orbit at the destination planet increases rapidly going in toward the Sun. At Mercury, the innermost planet, the velocity increment required to send a spacecraft into orbit is 32,000 ft/sec, added to 18,000 ft/sec for the initial transfer, giving a total of 50,000 ft/sec. Going away from the Sun, the velocity increment ΔV_2 increases to a maximum at about Saturn distance and then starts to decrease. This decrease is essentially caused by the diminishing gravitational attrac-

tion of the Sun at these great distances, requiring much lower planetary orbit velocities.

The Hohmann transfer, although requiring minimum energy, takes the longest time to reach the planets. Faster flights to the planets may be achieved by providing larger velocity increments than those shown here. However, a larger ΔV requires more propellant, and hence a smaller delivered payload. Figure 2 shows the variation of the required ΔV with flight time. The mission dates indicated are not necessarily proposed, but represent the dates of best encounter for the planets over the next 10 years. Higher velocity increments would be needed for other dates of opportunity. Shown in this plot are flyby probe and orbital missions to Mercury, Venus, Mars, and Jupiter. For each mission, the Hohmann transfer requirements are represented by the lowest velocity increment and the longest flight time. As flight time is decreased, the required velocity increment increases.
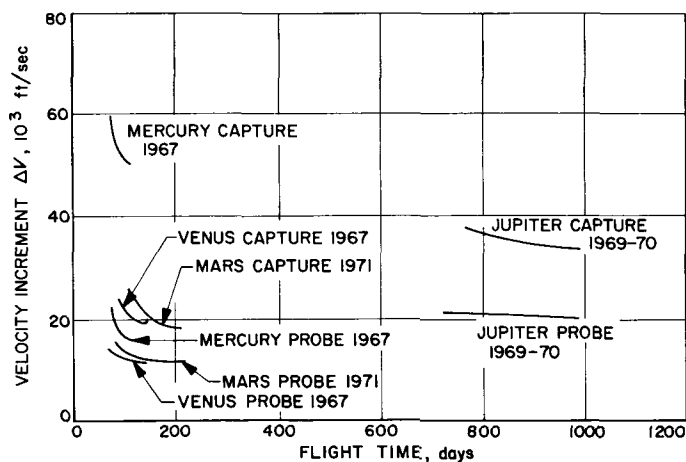

Fig. 2. Velocity requirements for planetary impulsive-transfer missions (best encounter)

The spacecraft weights that can be delivered to these planetary destinations are graphically portrayed in Fig. 3. This shows the performance capabilities of one- and two-stage chemical systems for which the specific impulse I_s is 430 sec and the structural factor λ_s $(M_p/M_p + M_s)$ is 0.85. Thus these curves represent the actual propulsion payload fractions. The two-stage system assumes that the first stage is used for heliocentric injection from low Earth orbit, and that the second stage is fired at the destination planet to achieve planetary orbit. For Mars

and Venus orbital missions, the chemical systems can yield payload fractions of 0.1 to 0.2. For the major planet probes, the two-stage chemical system can deliver payload fractions of 0.05 to 0.1. Payload fractions for the major planet orbital missions are extremely marginal.
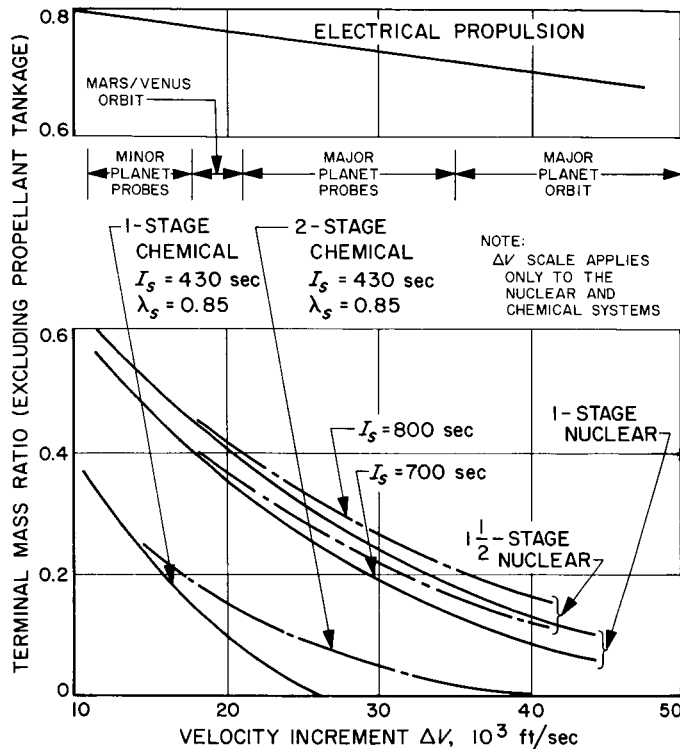

Fig. 3. Terminal mass fractions for planetary missions from Earth orbit

The performance capabilities of the nuclear-rocket systems with I_s of 700 and 800 sec are also shown in Fig. 3. Tankage was optimistically estimated at 10 percent of the propellant weight, since cryogenic hydrogen is required. But the weight of the nuclear-rocket system has not yet been subtracted from the terminal mass. The 1½-stage system drops empty propellant tanks after heliocentric injection from low Earth orbit, but the nuclear rocket is reused at the destination planet to provide the second velocity increment needed to go into planetary orbit. These curves also assume a true impulsive transfer. If thrust acceleration for nuclear rockets is less than approximately 0.5 g, additional losses would be involved. For Mars and Venus orbital missions, the nuclear rocket may deliver a terminal mass fraction of 0.40 to 0.45; a terminal mass fraction of 0.2 to 0.3 may be delivered at a major planet flyby. For major planet orbital missions, the nuclear rocket may deliver a terminal mass

fraction of 0.1. After the weight of the rocket system has been subtracted from the terminal mass, even the nuclear system appears marginal for the major planet orbital missions. A Mercury orbiter is beyond the capability of nuclear rockets if standard orbital transfers are considered.

The missions for electrical propulsion in Fig. 3 also show the terminal mass fraction that can be delivered to a planetary destination. Since hyperbolic excess velocity is shown only for planetary impulsive transfer with chemical and nuclear-rocket systems, electrical propulsion is compared on the basis of terminal mass for an equivalent planetary mission. The equivalent velocity increment is much higher for an electrical propulsion mission than for an impulsive transfer trajectory. A more convenient format for electrical propulsion missions is shown in Fig. 4, where mission dependence on spacecraft specific power level is clearly indicated. For major planet orbit missions, an electrical propulsion system may deliver a terminal mass fraction of 0.7.
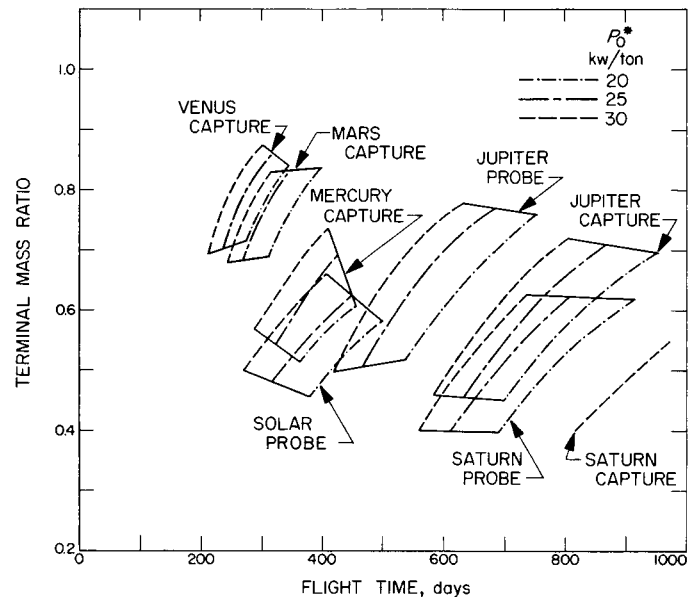

Fig. 4. Performance capabilities of electrically propelled spacecraft

A graphical display of terminal mass delivered to the planets, however, is not a true representation of the payload capabilities of the three systems shown. The propulsion system of the nuclear-rocket vehicle is included in the terminal mass but is no longer useful. The nuclear-electric supply of the nuclear-electric propulsion system, though operating, must also be charged mainly against the propulsion system. And initial Earth-orbit criteria must be introduced.

Considering the potential application of these systems to boost vehicles presently under NASA development, performance is compared in Fig. 5 for the *Saturn I-B*. An initial Earth-orbit spacecraft weight penalty has been introduced for electrical propulsion, starting with 20,000 lb in orbit rather than 30,000 lb. Allowing up to 5,000 lb for the spacecraft adapter, shrouding, and special startup equipment, a 700 n.mi. initial Earth orbit was chosen as a safe orbit for nuclear-electric system startup. Fifty percent (10,000 lb) of the initial spacecraft weight is then allotted for the nuclear-electric propulsion system. This includes the basic powerplant, shielding, power-conditioning and control equipment, and thrustors.
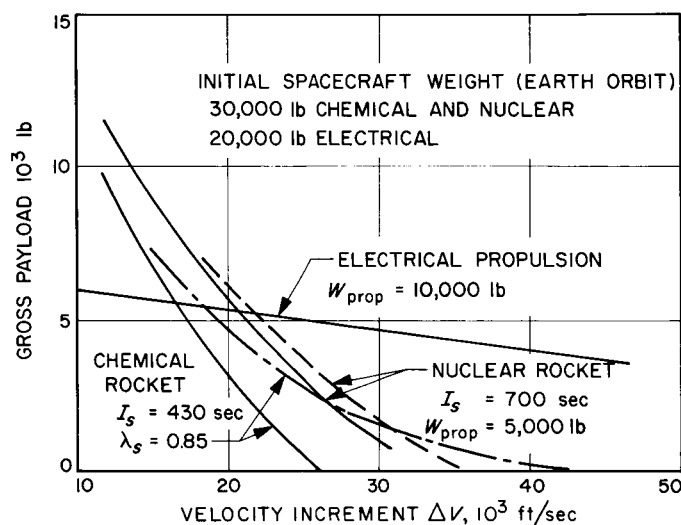

Fig. 5. Payload capabilities for chemical, nuclear, and electrical systems with *Saturn I-B* launch

A nuclear-rocket system with a specific impulse of 700 sec and a weight of only 5000 lb would barely become competitive with a two-stage chemical system.

Two tons or more of payload may be delivered anywhere within the solar system within fairly reasonable flight times by the electrically propelled spacecraft. Furthermore, it appears that all of the missions may be accomplished with a spacecraft of a single basic design. The inherent economy and improved reliability of such an approach is obvious.

The spacecraft power system is based on the SNAP-50/SPUR technology, operating at a power level up to 500 kwe. Approximately 80 percent would be delivered to the electric thrustors. The remainder would be consumed by the internal powerplant functions, by losses in power-conditioning and control, and by other power users on the spacecraft. The specific weight requirements of the propulsion system are in the range of 20 to 40 lb/kwe. The main constraint is to remain within the 10,000-lb weight limitation.

Flight time for the various planetary missions is worth further discussion. A comparison is made in Fig. 6 of flight times for electrical systems with those for chemical and nuclear systems. The right-hand edge of the curve for chemical and nuclear systems is the Hohmann transfer time, which may be reduced somewhat if a payload weight loss can be tolerated. At the lower end, the low-thrust and high-thrust curves cross over and are juxtaposed because of the electrical propulsion time lost during Earth spiral escape.
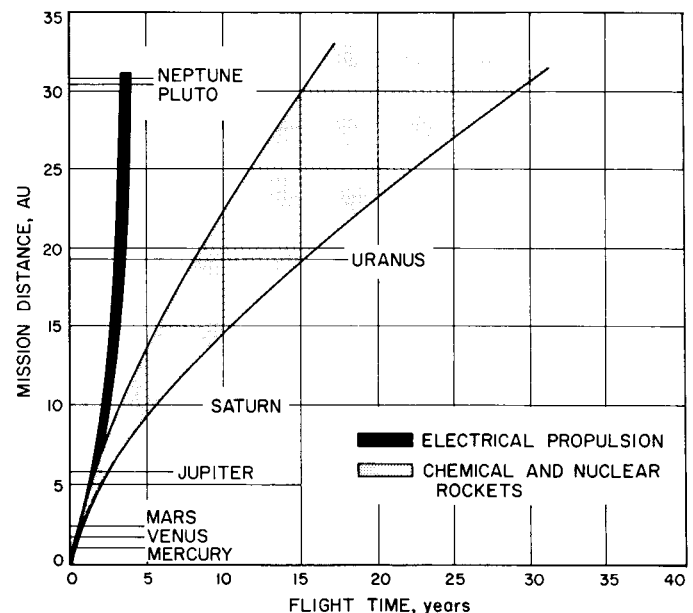

Fig. 6. Flight time requirements for planetary missions

It is interesting to note here that the planet Pluto, which has a 248-year orbit around the Sun, makes its closest approach to the Sun in 1984–85. Since Pluto would then be inside the orbit of Neptune, a space probe for rendezvous with Pluto at that time would be a significant scientific mission. But if a chemical or nuclear space probe were to be sent for that rendezvous, countdown at Cape Kennedy should probably begin today. On the other hand, the use of electrical propulsion would allow an additional 10 years of development time, and then five more years to build flight hardware.

Using the *Saturn V* launch vehicle, the missions for nuclear rockets look somewhat better (Fig. 7). A propul-

sion system weight of 10,000 lb is more realistic. Again, a weight penalty was imposed on the electrically propelled spacecraft to assure system startup in a safe Earth orbit. Power level required by propulsion is much higher, and electrical propulsion curves are indicated for propulsion system weights of 50,000 and 75,000 lb. For comparison purposes, the curve of the *Saturn I-B* electric spacecraft of Fig. 5 is also shown.
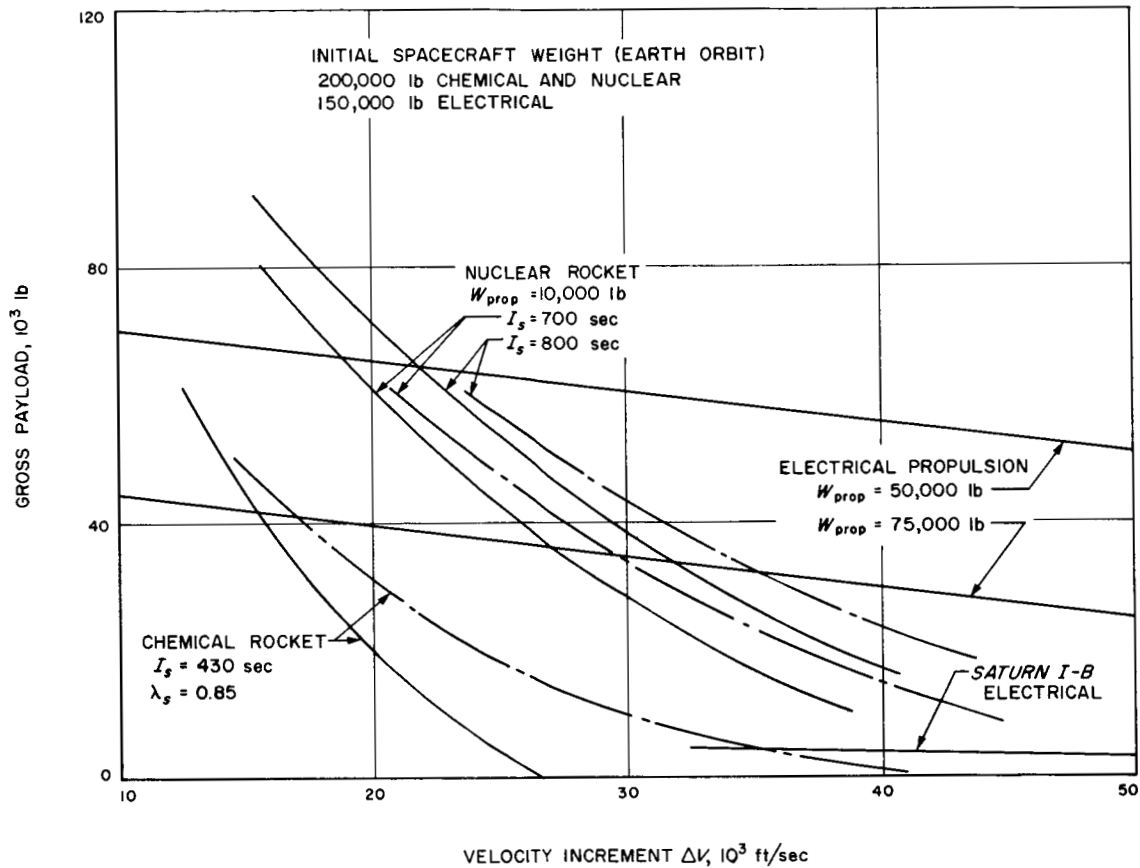

Fig. 7. Payload capabilities for chemical, nuclear, and electrical systems with *Saturn V* launch

IV. FUTURE GROWTH REQUIREMENTS

Eventually, larger boosters and more advanced propulsion systems will be developed. Nuclear propulsion systems may begin taking over the duties presently assigned to large chemical systems. The logistic support of lunar bases and the preparation for manned planetary expeditions may well consume most of our capacity in all forms of propulsion. High-performance boosters teamed with electrically propelled spaceships will spearhead man's conquest of the planets. But the extremely high energy propulsion required by the manned planetary program will still be exceeded by the needs of unmanned exploration. A single set of high-energy maneuvers will be required of the first-generation electrically propelled spacecraft; multiple sets of high-energy maneuvers will be required of the second generation. Thus a much greater versatility will be required for the electrical propulsion systems. Multiple start, variable thrust, and variable specific impulse are needed, while the same or higher efficiency must be maintained.

A comet chaser is an excellent example of a second-generation mission. Here, the spacecraft is required to make a rendezvous in space and, at the same time, to make an orbital plane change. Thereafter, additional maneuvers may be commanded to probe the tail of the comet for its constituents, temperatures, opaqueness, etc.

Halley's comet, the most prominent of the periodic comets, is due to return to the vicinity of the Sun in the 1985–86 time period. However, of all the periodic comets it is the most difficult target since it is the only one with an orbital inclination exceeding 90 deg; the others move in the same direction as the planets, and 35 out of the 40 known have orbital planes tilted at less than 45 deg to the ecliptic. Jupiter's apparent dominance over most of the cometary trajectories might also be investigated as a part of a second-generation electrical spacecraft mission.

The exploration of the asteroids is another possible type of high-energy-spacecraft mission, while rendezvous with the moons of the various planets also represent multiple maneuver requirements. An orbital spacecraft that sends down small landing craft would be typical of second-generation electrical spacecraft. We may expect that such vehicles will be phased into the planetary exploration program toward the end of the 1970's. Whether sufficient capability to rendezvous with Halley's comet will have been developed is rapidly becoming a point of serious conjecture. If not, we must be satisfied with easier tasks. It is conclusively evident that spacecraft utilizing nuclear-electric propulsion systems will play a commanding role in planetary explorations of the future.

NOMENCLATURE

I_s specific impulse, sec

M_p propellant mass

M_s structural mass

P_0^* specific power level, kwe/ton

ΔV total velocity increment, ft/sec

ΔV_1 velocity increment required to send a spacecraft from an initial, low Earth orbit to achieve a flyby (probe), ft/sec

ΔV_2 velocity increment required for a spacecraft to achieve a capture orbit at the destination planet, ft/sec

W_{prop} propulsion system weight, lb

λ_s structural factor $= \dfrac{M_p}{M_p + M_s}$

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